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Summary

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The Lunar Orbiter is an unmanned lunar reconnaissance satellite which will take high-resolution photographs of large areas of the lunar surface to assist in selecting a landing site for Apollo. Because of various mission aspects and the fact that the Orbiter employs a film-type camera, the requirements placed on the telecommunications system are different than with previous U.S. spacecraft.

This paper describes the design of both the spacecraft and ground equipment of the Lunar Orbiter Telecommunications System. Some of the unique features of the design include the use of vestigial sideband modulation for transmission of the picture data, a full-verification command system, a low-gain antenna system whose pattern approaches that of an isotropic radiator over a large portion of the radiation sphere, and a dual-power-level transmission system which does not use RF switching. Included in the paper is a link-design chart which shows the S/N ratios and performance margins obtained for all operating modes.

Introduction

The primary purpose of the Lunar Orbiter program is to obtain lunar topographic data to assist in selecting an Apollo landing site. It will accomplish this by taking high-resolution photographs from lunar orbit at an altitude of about 25 miles. Because of the moon's rotation, the Lunar Orbiter will overfly different areas of the lunar surface on successive orbits, enabling a single spacecraft to photograph a number of different potential Apollo landing sites. The photographs will be obtained using a film-type camera, the pictures developed onboard, and then scanned and transmitted to earth.

Tracking of the Lunar Orbiter will yield information concerning the size and shape of the moon and its gravitational field. In addition, the spacecraft will carry a limited number of sensors which will provide some data on the lunar environment.

The concept and design of the Telecommunications System described in this paper were accomplished by the prime Lunar Orbiter contractor, The Boeing Company, and its major subcontractor, RCA. The Project is being managed by the NASA Langley Research Center.

Mission Plan

The first Lunar Orbiter is scheduled for launch in 1966, using an Atlas-Agena vehicle. Figure 1 shows the overall telecommunications and

tracking approach which is similar to that employed for Ranger and other lunar and planetary spacecraft. During the launch phase, telemetry and tracking will be accomplished by the Eastern Test Range (ETR) stations. At about the time the spacecraft is inserted into a translunar trajectory, tracking, telemetry, and command responsibility will be assumed by the Goldstone, Madrid, and Woomera Stations of the NASA Deep Space Instrumentation Facility (DSIF). Centralized control of the mission will be accomplished from the NASA Space Flight Operations Facility (SFOF) in Pasadena, California.

During the launch phase, telemetry data from the spacecraft will be available over two RF links: the spacecraft S-band link and the Agena VHF link. Transmission over the Agena VHF link is accomplished by modulating one of the Agena IRIG subcarriers with the spacecraft PCM bit-train. After separation from the Agena, communications with the spacecraft will be at S-band frequencies only, the down-link and up-link frequencies being in the 2290 to 2300 Mc and 2110 to 2120 Mc International Space Bands, respectively.

Spacecraft Configuration

Figure 2 shows the general configuration of the 850-pound spacecraft which is about $5\frac{1}{2}$ feet high and 5 feet in diameter, excluding the solar panels and antennas. The span across the deployed antenna booms is about $18\frac{1}{2}$ feet.

With the exception of the rocket engine and its fuel tanks, which are used for midcourse maneuvers and deboost into lunar orbit, essentially all of the major spacecraft components are attached to an equipment mounting plate. That mounting plate, the underside of which remains oriented toward the sun at all times except when the spacecraft is maneuvering, provides thermal control for the components attached to it.

In the flight configuration, all of the main spacecraft structure above the equipment mounting plate is covered with a highly reflective shroud of aluminum-coated mylar which constitutes part of the thermal control system. The only protrusions through that shroud are the camera lenses and micrometeoroid detectors.

The spacecraft power system is a conventional solar array/storage battery type. A 12 ampere-hour nickel cadmium battery will be used to supply the spacecraft power requirements during the launch phase prior to solar array deployment and during those periods of the lunar orbit when the spacecraft is in the moon's shadow. Because of the

rather severe weight penalty for battery capacity, it has been imperative to minimize the power consumption of the "residual" power loads, including such communication system items as the transponder, command decoder, etc. When the solar array is in sunlight a maximum of about 375 watts is available to handle all power demands including battery recharging. The voltage of the spacecraft dc-power bus can vary from a minimum of 22 volts to a maximum of 31 volts when the array is in operation. A shunt regulator is used to prevent the voltage from exceeding 31 volts.

The star Canopus and the sun are the primary references for spacecraft attitude orientation. For maneuvering, or when those references are occulted, a strapped-down gyro system is used. Attitude control is accomplished by a cold-gas system.

The spacecraft camera system employs two lenses which take simultaneous pictures on a roll of 70-mm-wide aerial film. One of the lenses has a 24-inch focal length and can take pictures from an altitude of 46 km with a resolution of about 1 meter. The other lens, which has a focal length of about 3 inches, will take pictures with a resolution of about 8 meters. The film is developed onboard using a method which presses the film into contact with a web that contains a single-solution processing chemical. After the film has been dried, it is ready for readout and transmission to the ground.

Figure 3 is a drawing of the readout system. The light source for film scanning is a line-scan tube. The tube itself sweeps the beam of light through an excursion of 0.1 of an inch in a direction parallel to the film travel. An optical system sweeps the beam of light across the film in a transverse direction and also serves to focus that spot of light. As the figure shows, scanning with sweeps of 0.1 inch begins at one edge of the film and continues across the film until the opposite edge is reached. At that time the film is advanced 0.1 inch and the film is scanned in the opposite direction. This sequence is repeated, with the scan rates being such that 43 minutes are required to scan about 1 foot of film which contains one high-resolution and one medium-resolution photograph. Collecting optics direct the transmitted light into a photomultiplier, and the resulting electrical signal is conditioned and mixed with synch and blanking pulses and fed to the communication system modulator.

The spacecraft carries a film supply of about 200 feet which is sufficient to photograph, from an altitude of 46 km, 12,000 sq km of lunar surface with a resolution of 1 meter, and 200,000 sq km with a resolution of 8 meters. It is interesting to note that about one million standard commercial TV pictures would be required to photograph that area with comparable resolution.

The spacecraft "brain" is a programmer which accepts inputs from the earth via the command system. That programmer is essentially a digital data

processing system and will control about 65 functions within the spacecraft.

The only major spacecraft system not described above is the telecommunications system which will be discussed in detail in the remainder of this paper.

Telecommunication System Design Objectives and Constraints

The basic functions assigned to the Lunar Orbiter communication system include data transmission (telemetry and video data), command reception, and measurement of Doppler and range.

The design of a system to satisfy those basic functional requirements was constrained by a number of factors, including the following:

- (1) Compatibility with the DSIF Stations.
- (2) Each RF link to have a nominal performance margin of not less than 6 dB.
- (3) Video (photographic data) to be transmitted in analog form.
- (4) Video (data rate) to be approximately $1/4$ megacycle.
- (5) Ability to transmit telemetry data during video transmission.
- (6) Provide a video data output S/N ratio of not less than 33 dB, p-p/rms.
- (7) Provide a command link having a bit error rate not greater than 10^{-5} .
- (8) Provide a telemetry bit error rate not greater than 10^{-5} .
- (9) Provide a spacecraft antenna system which, at lunar distance, would enable command capability and telemetry reception for any spacecraft orientation.
- (10) Have an overall reliability of not less than 95 percent for the first 30 days of the mission.
- (11) Use existing, space-qualified, hardware components wherever possible.
- (12) Provide for on-off control from the earth for all spacecraft RF emissions.
- (13) Minimize system power consumption.
- (14) Sideband energy outside of $3\frac{1}{3}$ Mc channel bandwidth to be at least 30 dB down.
- (15) Avoid the use of RF power switches in the spacecraft.
- (16) Provide for two-way Doppler tracking.

(17) Provide for ranging measurements using the JPL Pseudo Noise ranging system.

The following sections of this paper describe the design concepts and hardware employed to satisfy these objectives and constraints.

Modulation Techniques

For each of the telecommunication links and emission modes, various modulation techniques were carefully considered. The emission mode which required the greatest amount of consideration was the one involving transmission of the video data.

Two video-data modulation techniques were considered: FM/PM and VSB-AM/PM. Vestigial sideband-amplitude modulation (VSB-AM) was chosen primarily on the basis of containing more of the radiated energy within the allotted 3.33-Mc spaceband channel. That consideration was imposed as a design constraint due to the possibility of other spacecraft operating in an adjacent channel. When compared to the FM/PM technique considered, the VSB system also offered a slightly higher output S/N ratio, although at the expense of a slightly higher threshold.

The video data, which occupy a frequency spectrum from 0 to 230 kc, are first double-sideband, suppressed-carrier modulated onto a 310-kc subcarrier, and then the upper sideband removed by filtering. A 38.75-kc pilot tone is derived from the 310-kc subcarrier oscillator and is transmitted for use by the ground demodulator for subcarrier reinsertion. The 50 bps PCM telemetry data are diphase modulated (0° or 180°) onto a 30-kc subcarrier. The video, pilot tone, and telemetry signals are then summed, and the resulting composite signal phase modulates the S-band carrier. The frequency spectrums of the video and composite signals are shown in figure 4. The effective modulation indices are: video - 3.6 radians peak, pilot tone - 0.47 radian peak, and telemetry - 0.2 radian peak. When transmitting video data, the spacecraft operates on a 10-watt transmitter and a directional antenna with a gain of 23.5 dB. This insures that the FMFB ground demodulator operates above threshold. The PM noise improvement factor of 17.8 dB provides adequate signal-to-noise ratio at the video output of the communications system.

When video is not being transmitted, the spacecraft operates on a 0.5-watt transmitter to conserve battery power. Doppler tracking, ranging, and telemetry transmission can be accomplished in this low-power mode using the low-gain antenna. The same 50 bps PCM telemetry data are telemetered to earth on a 30-kc subcarrier as during video transmission. However, while operating in the low power mode, the modulation index is increased from 0.2 radian peak to 1.5 radians peak to enhance the output S/N ratio at the receiving station. In all cases the telemetry bit error rate is expected to be 10^{-5} or less. Ranging is accomplished using the JPL turnaround ranging system. The pseudo noise ranging code is phase modulated onto the

up-link S-band carrier with an index of 1.2 radians peak. After detection in the transponder, the code is then remodulated on the down-link carrier with an index of 0.41 radian peak and is transmitted simultaneously with telemetry.

Several subcarriers are used in the command transmission system. The digital command word is frequency-shift-keyed onto one of the subcarriers. All of the subcarriers utilized are phase modulated on the up-link carrier, each with an index of 0.7 radian peak. The frequencies of the subcarriers and their sequence of transmission are classified for security purposes. To insure that commands are received correctly, the command words are telemetered back to the DSIF via one of the data slots in the 50 bps telemetry frame before the commands are executed by the spacecraft. Since commands are received by the low-gain antenna, command capability exists for virtually any spacecraft orientation.

Spacecraft System Description

The spacecraft telecommunication system block diagram, figure 5, shows how certain hardware constraints and requirements are satisfied, and the theoretical design approach is implemented. Except for the periods of video transmission, the spacecraft will operate with the transponder exciter and low-gain antenna. For video transmission, the spacecraft effective radiated power will be increased by a TWT RF amplifier and high-gain antenna.

The appropriate spacecraft emission mode will be selected by earth command: (1) telemetry and ranging, (2) video and telemetry, or (3) telemetry alone. In addition to turning "on" and "off" the transponder ranging module and TWT RF amplifier as required for the various emission modes, the command system will also have the capability to completely silence the spacecraft by turning "off" the exciter section of the transponder.

Low-Gain Antenna

The low-gain antenna pictured in figure 6 is a biconical-diskcone antenna mounted at the end of an 82-inch boom. It radiates an elliptically polarized wave in the TEM mode and is fed by the boom which serves as a low-loss coaxial transmission line. The VSWR is less than 1.6:1 for the 2290- to 2300-Mc band and less than 3.0:1 for the 2110- to 2120-Mc band.

During the launch phase when the spacecraft is enclosed by a heat protective shroud, the antenna is stowed using a hinge mechanism located at the interface between the boom and spacecraft. After shroud ejection, the antenna is deployed by a squib-initiated, spring-driven system which incorporates a latching mechanism for locking the boom in the deployed position.

Of particular significance is the closeness with which the low-gain antenna approximates an isotropic radiator. Figure 7 is a radiation

pattern of the antenna which shows a gain of 0 dB in the maximum radiating plane, XZ, and -3 and -6 dB beamwidths of 60° and 120°, respectively. The antenna is useful for transmitting telemetry over 95 percent of its radiation sphere and is capable of receiving commands over 99 percent of the radiation sphere. Figure 8 shows that the antenna boom is oriented approximately perpendicular to the ecliptic plane when the spacecraft attitude control system is referenced to the sun and Canopus; therefore, the earth will lie in the maximum radiation plane of the antenna.

High-Gain Antenna

The high-gain antenna is a 36-inch parabolic dish of lightweight honeycomb construction. It is attached to the end of a 52-inch boom which acts as a coaxial transmission line. The antenna is deployed from its stowed position at spacecraft shroud ejection in the same manner as the low-gain antenna. Some of the pertinent characteristics of this antenna are:

GAIN:	23.5 dB
BEAMWIDTH:	10°
EFFICIENCY:	61 percent
BANDWIDTH:	2290 to 2300 Mc
FEED TYPE:	Turnstile with cup-shaped reflector
FOCAL LENGTH:	12 inches
POLARIZATION:	Right-hand elliptical
AXIAL RATIO:	<1.59:1
VSWR:	<1.3:1
WEIGHT:	2.3 lb (dish and feed)

An unusual feature of the Lunar Orbiter use of this antenna is the method of pointing it toward the earth using rotation about only one axis. Figure 8 shows that when the attitude reference system is locked on the sun and Canopus the antenna boom is approximately perpendicular to the ecliptic plane. Therefore, as the moon traverses its orbit, it is necessary only to rotate the antenna about the boom axis to direct the pattern toward the earth. Rotation is accomplished using a stepping motor and gear train located between the base of the boom and the spacecraft. The motor and gear train permits stepping the antenna in 1° increments and will be controlled from earth by command. A readout of the antenna position is accomplished using a digital angle encoder.

Ability to direct the antenna at the earth by rotating it about only one axis is dependent upon certain mechanical and geometrical factors. The mechanical factors, such as manufacturing tolerances, deployment tolerances, positioning resolution, and attitude-control deadband and limit cycles, have been minimized by design and

construction. However, there are several geometrical factors that have a significant effect. The 5° tilt of the earth-moon plane with respect to the ecliptic will cause a tilt in the spacecraft roll axis as the spacecraft goes alternately above and below the ecliptic. A more serious geometric error is caused by spacecraft movement about the roll axis due to the 14° off-ecliptic-normal location of Canopus.

By proper offsetting of the star tracker and shimming the antenna hinge mechanism for a particular launch period, the geometrical errors can be minimized. By such techniques, the sum of the mechanical and geometrical pointing errors can be kept to less than one-half the antenna beamwidth for a given 30-day period. This approach is satisfactory since the video transmission, which requires the use of the high-gain antenna, will be completed within 30 days.

Modulation Selector

The modulation selector, shown in figure 9, contains the subcarrier oscillators and other circuits required to develop the modulation spectrums and control the spacecraft emission modes. The emission modes, either telemetry only or telemetry and video, are selected by earth command.

In the telemetry-only mode, the 50 bps PCM bit train is diphasic modulated on the 30-kc subcarrier by the double-balanced modulator such that a 180° phase change takes place only when the NRZ bit train changes state. The double-balanced modulator used is basically a ring modulator utilizing a matched diode quad for the choppers. The modulated output is a double sideband-suppressed carrier spectrum. In order to meet stability requirements and maintain the modulation index at its proper value, the modulated wave is limited before being filtered by the 30-kc bandpass filter.

The video data, which comes from the photo readout system as a 0- to 230-kc video baseband, is amplitude modulated on the 310-kc crystal controlled subcarrier in the double-balanced modulator resulting in a double sideband-suppressed carrier modulated output. The ring modulator is similar to the one discussed above for the telemetry case and accomplishes greater than 40-dB carrier suppression. The upper sideband of the modulated wave is removed by the vestigial filter which is a fifth order modified Butterworth with a linear phase response. The filter is designed to have a low-frequency cutoff beginning at 80 kc with a response at 40 kc which is 25 dB down from that at 80 kc. The high-frequency amplitude response of the filter is designed to be skew-symmetrical about 310 kc with a response 6 dB down at that frequency.

The video modulated wave thus developed is mixed in the summing amplifier with a 38.75-kc pilot tone, which is coherently related to the 310 kc by the divide-by-eight multivibrator circuit, and with the telemetry modulated 30-kc subcarrier to form the spectrum discussed above and shown in figure 4.

Transponder

The Lunar Orbiter transponder is basically an extension of the coherent transponder used on Mariner 4 and makes use of the standard modules developed by JPL. An RF diplexer, containing a directional coupler, a power monitor and filters, has been incorporated in the Orbiter transponder in order to couple some of the exciter power to the TWT RF amplifier, perform power output monitoring, and filter the transponder RF inputs and outputs. In addition, a power converter has been included in the Orbiter transponder so that it can operate directly from the spacecraft 28-volt dc power bus. Switching provisions have been incorporated so that the exciter section of the transponder and the ranging module can be turned "on" and "off" by earth command.

The transponder will operate in a noncoherent mode using a self-contained crystal-controlled auxiliary oscillator until an up-link carrier is received with sufficient signal strength to switch to the coherent VCO. Coherent operation will translate the up-link carrier frequency by the exact ratio 240/221 for retransmission.

For lunar distances, it was found that the loop noise bandwidth of the transponder could be increased from the 20 cps required by Mariner to 100 cps and still have a receiving threshold which would allow the communications link to perform as planned. By increasing the loop noise bandwidth to 100 cps, the capability to coherently track greater Doppler rates is provided, allowing two-way Doppler tracking during the worst-case spacecraft accelerations.

Other transponder characteristics are:

Threshold:	-142 dBm
Noise figure:	12 dB
Exciter power output:	500 mw
Modulation sensitivity:	2 rad/volt
Weight:	12.75 lb
Power consumption:	Less than 17 watts

TWT RF Amplifier

The Lunar Orbiter TWT RF amplifier uses an existing space-proven tube, the 349H. The prime reason for selecting this device rather than another power amplifier, such as a triode cavity or amplatron, was its proven long life.

The RF amplifier package consists of the 349H, a dc-dc power converter and an RF bandpass filter-power monitor. The filter has 79 dB of rejection at the transponder receiver frequency, a matched passband of approximately 40 Mc at the transmit frequency with an insertion loss of 0.35 dB, and greater than 60-dB attenuation at the second and third harmonics. The filter characteristics are accomplished by combining a band reject filter with

a low pass filter. Some of the other characteristics of the TWT RF amplifier are:

POWER OUTPUT:	10-13 watts
RF GAIN:	27 dB
NOISE FIGURE:	38 dB
OVERALL EFFICIENCY:	24 percent
D-C POWER REQUIRED:	54 watts, maximum
WEIGHT:	5.5 lb

Command Decoder

The command decoder contains the demodulators for the command subcarriers, the command-data shift registers, and other circuits used for address recognition and code verification.

When a command is received, it is temporarily stored in the data shift registers so that the multiplexer encoder can nondestructively sample the command word and telemeter it to earth for verification. The 50 bps clock required for read-out of the shift registers is supplied by the multiplexer encoder. Upon verification on earth that the correct command is being held in the registers, the command word is transferred to the spacecraft programmer to be acted upon immediately or to be stored and acted upon at a designated later time. If a wrong command is being held, the register can be cleared and the command retransmitted.

The first 4 bits of the command word make up an address code allowing multiple spacecraft operation; only upon recognition of a correct address will the address verification register permit the remainder of the command word to be fed to the data shift register. The remaining bits in the 26-bit command word contain the command message.

The command transmission rate on the up-link carrier is 20 bps. The bit stream is read from the decoder into the programmer at a 20 bps rate controlled by an internal 20 bps generator which is corrected by a clock signal from the spacecraft programmer.

Multiplexer Encoder

The multiplexer encoder, figure 10, is essentially a standard PCM encoder consisting of analog and digital multiplexers, an analog to digital converter (ADC), and several registers. It is unusual only with respect to the type of ADC employed and the manner in which it is interconnected to the command decoder and spacecraft programmer.

The encoder converts analog and digital input data into a 50 bps serial NRZ bit train. Each frame is equivalent to 128 9-bit words, thereby providing one complete frame every 23 seconds. Each frame contains 78 analog

channels, 127 1-bit "event" channels, 9 digital channels and a 43-bit Legendre frame synch word.

The fundamental clock rate of 50 bps for the encoder is developed by the spacecraft programmer; however, an encoder sensing circuit will transfer operation to an internal free-running oscillator should the programmer clock fail. All "shift" and "load" pulses used for timing and operation of the encoder are internally generated and the timing is based on the fundamental 50 bps frequency.

The Lunar Orbiter encoder deviates from standard designs by employing a closed-loop voltage-controlled oscillator (VCO) in the ramp-type ADC in place of the usual crystal oscillator. The frequency determining voltage for the VCO is developed across a relatively large capacitor which is connected either to a current sink or a current source. A precision voltage, which is set to encode to a specified digital count, is encoded sequentially with the data samples. A circuit which senses the count developed by that voltage will, if the count is either above or below that specified, connect the VCO capacitor to either decrease or increase the frequency. Consequently, the VCO frequency is constantly adjusted so that the precision voltage and the input analog data are encoded correctly.

In order to telemeter data from the spacecraft programmer, it has been necessary to provide a storage register in the multiplexer encoder because of the difference in programmer and multiplexer encoder data rates. Transfer of programmer data into that 20-bit storage register is initiated by a load signal from the multiplexer encoder.

The digital multiplexer circuit in the encoder is used to assemble the digital words from the ADC storage register, the 20-bit storage register, the frame synch generator, and the 1-bit "event" channels in the proper frame sequence. A 9-bit output register then adds the complementary bit to each group of 8 bits and also inserts the command word from the command decoder storage register into the serial bit train when the command decoder sends a "register full" signal.

Ground System Description

The Lunar Orbiter will use the Deep Space Stations at Goldstone, Madrid, and Woomera. The equipment configuration employed at those stations consists of both standard station equipment and special Lunar Orbiter project equipment.

When the spacecraft operates in the low-power mode, Doppler tracking and carrier demodulation is accomplished by the DSIF phase-lock receiver (see the ground system block diagram, fig. 11). The 30-kc telemetry subcarrier is taken from the telemetry output of the DSIF receiver and is then demodulated in the 30-kc telemetry demodulator. After detection and decommutation, the PCM bit train is entered into an SDS-920 computer. The computer edits the telemetry data, inserts GMT,

and formats the data for transmission to the SFOF via high-speed-data and teletype lines.

During transmission of video data, the DSIF receiver operates in a noncoherent mode since the modulation index of 3.6 radians often produces phase reversal of the carrier and prevents carrier tracking. The receiver gain and VCO frequency are controlled manually during this type operation. The 10-Mc IF output of the receiver is demodulated by an FMFB demodulator. Coherent detection of the 310-kc video subcarrier takes place in the video subcarrier detector, from whence the video data are fed to the video data photographic recorder where the pictures are reproduced. After the telemetry subcarrier is separated from the composite signal by filtering, the telemetry data follow the same path described in the previous paragraph for low-power operation.

In addition to the video data being recorded photographically, it is also recorded on a pre-detection tape recorder. Post-detection magnetic tape recording is used when only telemetry is being received from the spacecraft.

Deep Space Instrumentation Facility

Since the DSIF has been described previously in the literature, only a brief summary of its characteristics will be given here.

The transmitter produces up to 10 kW of S-band power and can be phase modulated up to 3.0 radians peak. The phase-lock, double-superheterodyne receiver has a traveling wave maser front end with an equivalent noise temperature of 55° Kelvin. The threshold loop noise bandwidth ($2 B_{LO}$) of the carrier tracking loop is 12 cps.

Both transmitting and receiving are accomplished using an 85-foot-diameter, Cassegrainian fed, parabolic antenna with a gain of 53 dB. When pointed at the Lunar disk, the antenna adds another 110° K to the system noise temperature.

FMFB Demodulator

When receiving video data, an FMFB demodulator is used instead of a standard FM discriminator because of its threshold improvement characteristics. The demodulator thresholds at a C/N ratio of 7.0 dB measured in a 3.3-Mc bandwidth. It utilizes an internal IF of 45 Mc with a feedback factor of 9.5 dB, an IF noise bandwidth of 1.0 Mc, and a loop noise bandwidth of 600 kc. The device incorporates its own AGC circuit and for improved linearity employs two VCO's operating in push-pull.

Video Subcarrier Detector

A block diagram of the video subcarrier detector is found in figure 12. The pilot tone is used to recover the 310-kc subcarrier in a phase-locked loop. The loop employs a 100-cps noise bandwidth during acquisition, and after

lockup the bandwidth is switched to 1 cps to minimize phase jitter on the reconstructed 310-kc subcarrier. An active phase shifter is used to equalize the variation of phase response with frequency exhibited by the bandpass filter. This is necessary to maintain the proper phase relation between the reconstructed and the original subcarriers. The modulated video subcarrier is integrated prior to detection to compensate for the effect of an FM discriminator when used to demodulate a FM signal. The subcarrier then is demodulated in a product detector. The group delay of the entire communications system is kept constant within $\pm 0.8 \mu\text{sec}$ over the video frequency range of 0 to 230 kc by a phase equalizer in the output stages of the video subcarrier detector. The amplitude response through the entire communication system is flat within $\pm 0.5 \text{ dB}$ from 0 to 200 kc with a gradual roll-off to -2 dB at 230 kc. The video subcarrier detector also contains a bandpass filter which separates the 30-kc telemetry subcarrier for further processing.

Telemetry Equipment

The 30-kc demodulator employs a product detector to demodulate the telemetry subcarrier. The 30-kc subcarrier is recovered using a frequency doubling technique and a phase-locked loop having a threshold S/N ratio of 8.8 dB measured in the threshold loop noise bandwidth of 10 cps. Bit detection is accomplished with a reset integrator, and bit synchronization is maintained with a phase-locked loop. The telemetry demodulator detects the 43-bit frame synchronization word, and separates the PCM data into 128 9-bit words. In addition to supplying the demodulated data to the computer, the demodulator also feeds a printer that continuously displays all telemetry channels and a Nixie device capable of displaying any channel of information, one channel at a time.

Command Equipment

The command equipment normally accepts commands from the SFOF via the on-site computer. This process will be described in a later section of this paper. If a failure occurs in the SFOF-DSIF command link, commands can also be entered into the equipment using a paper tape reader or with manual switches.

The command words are frequency modulated onto a subcarrier and then mixed with other command system subcarriers. This combined signal is then fed to the DSIF transmitter modulator, where it is phase modulated onto the S-band carrier. When the command word is returned via telemetry from the spacecraft, it is routed into the command equipment. There it is automatically compared with the transmitted word, and, if the two are identical, an execute command is sent to the spacecraft, completing the command cycle.

Self-Test Equipment

Sufficient test equipment is provided to checkout all project ground equipment in the video, telemetry, and command links. The test

equipment is capable of isolating failures down to the component level by inserting appropriate test and simulated data signals into various points in the system.

A test transponder with electrical characteristics identical to the spacecraft transponder can be modulated by simulated video, telemetry, or ranging data, and its S-band output fed into the front end of the DSIF receiver.

A special purpose signal generator produces a signal at the 10-Mc IF frequency for insertion into the FMFB demodulator. This IF signal can be modulated by video and telemetry and can be mixed with noise. The video and telemetry signals used to modulate the test transponder and signal generator can also be fed directly into the video subcarrier detector and the 30-kc telemetry demodulator.

SFOF Data System

The Lunar Orbiter mission will be centrally controlled from the Space Flight Operations Facility in Pasadena, California. Located in the mission analysis and control areas of that facility will be the engineers whose responsibility is to direct the flight mission. They will have available spacecraft telemetry and tracking data and the results of analysis programs to determine the status of the spacecraft and its trajectory. Their control of the spacecraft will be by means of command messages originated at that facility and relayed to the remote DSIF stations for transmission to the spacecraft. Figure 13 is a data flow diagram of the SFOF system.

Telemetry data from the spacecraft will be received at the SFOF in real-time via a high-speed data line. In the event of an outage of that line, the telemetry data will be transmitted over TTY lines. Tracking data will be handled using TTY circuits.

At the SFOF, a 7288 serial/parallel converter is used as a buffer between the communication circuits and the 7040 computer. The raw telemetry and tracking data will be inserted on the 7094 storage disk by the 7040 with instructions to convert the data to engineering units. After the 7094 has processed the data it will be returned to the disk and the 7040 instructed to route the data for display in the mission control areas. Raw and converted telemetry and tracking data are recorded on magnetic tape at several key points in the data flow system along with paper tape records of the teletype line data to insure that little or no information will be lost due to on-line equipment failures.

Commands for the spacecraft are inserted into the input/output consoles in the mission control areas by either punched cards or typewritten coded English. The 7040 will store the command on the 7094 storage disk with instructions to convert the command to binary form. The 7040 will take the converted command from the disk and transmit it over teletype lines to the on-site computer in

triplicate. The on-site computer will either verify receipt of three identical command messages or an interrogation and verification cycle will continue until correct message receipt has been verified. Once a message has been received it will be sent immediately to the command generation equipment described previously, or held in storage for transmission at a later time. As a back-up mode, commands can be inserted directly on the teletype line using a paper tape reader and a manually punched tape.

The 7094 computer has also been programed to perform certain mission analysis functions, such as spacecraft trajectory determination, spacecraft thermal management, etc. In that case, the previous data which have been stored on the disk will be used, and the results of the analysis programs displayed either on strip charts or plot boards in the mission control areas.

Communications Link Design

Shown in table I is a summary of the Lunar Orbiter Communications link design for operation at lunar distance. The analysis is based on a system noise temperature of 165° K for the DSIF receiver and 3400° K for the transponder receiver. The thresholds for carrier tracking are the signal-to-noise ratio which produce a 30° rms phase jitter in the phase-lock loop. The minimum margin above threshold is 6.7 dB in the video link. With a PM noise improvement factor of 17.8 dB, the output signal-to-noise ratio for the video data is 40.5-dB peak-to-peak/rms in an output bandwidth of 230 kc, assuming a noiseless input signal from the spacecraft photo subsystem. Normally the signal from the photo subsystem has a signal-to-noise ratio of 24-dB peak-to-peak/rms; thus, the communications system noise has little effect on the noise content of the reconstructed pictures. Though not shown in the table, in the high-power mode the telemetry output S/N ratio is 44.9-dB rms/rms in a 50-cps bandwidth, and the pilot tone S/N ratio is 51.9-dB rms/rms in a 1-cps bandwidth.

Concluding Remarks

At this writing, preliminary tests have been conducted which show that the Lunar Orbiter Telecommunication System essentially satisfies the design requirements and performs as intended. Prior to the first launch, more extensive testing will have been completed and will serve further

to evaluate performance of the Telecommunication System at both the system and spacecraft level. Those tests will include operation of a spacecraft through two simulated mission cycles in a thermal-vacuum chamber as well as tests at the Goldstone DSIF to verify compatibility of the spacecraft and ground RF systems.

As a result of one of the system design objectives - utilization of already developed and qualified components wherever possible - an additional degree of confidence has been gained from Mariner 4's success, since the Orbiter transponder and RF amplifier are closely similar to the Mariner units.

Therefore, it is believed that the Lunar Orbiter Telecommunication System will, by virtue of design approach, performance margins, hardware selection and test results, satisfactorily fulfill its role in the Lunar Orbiter mission.

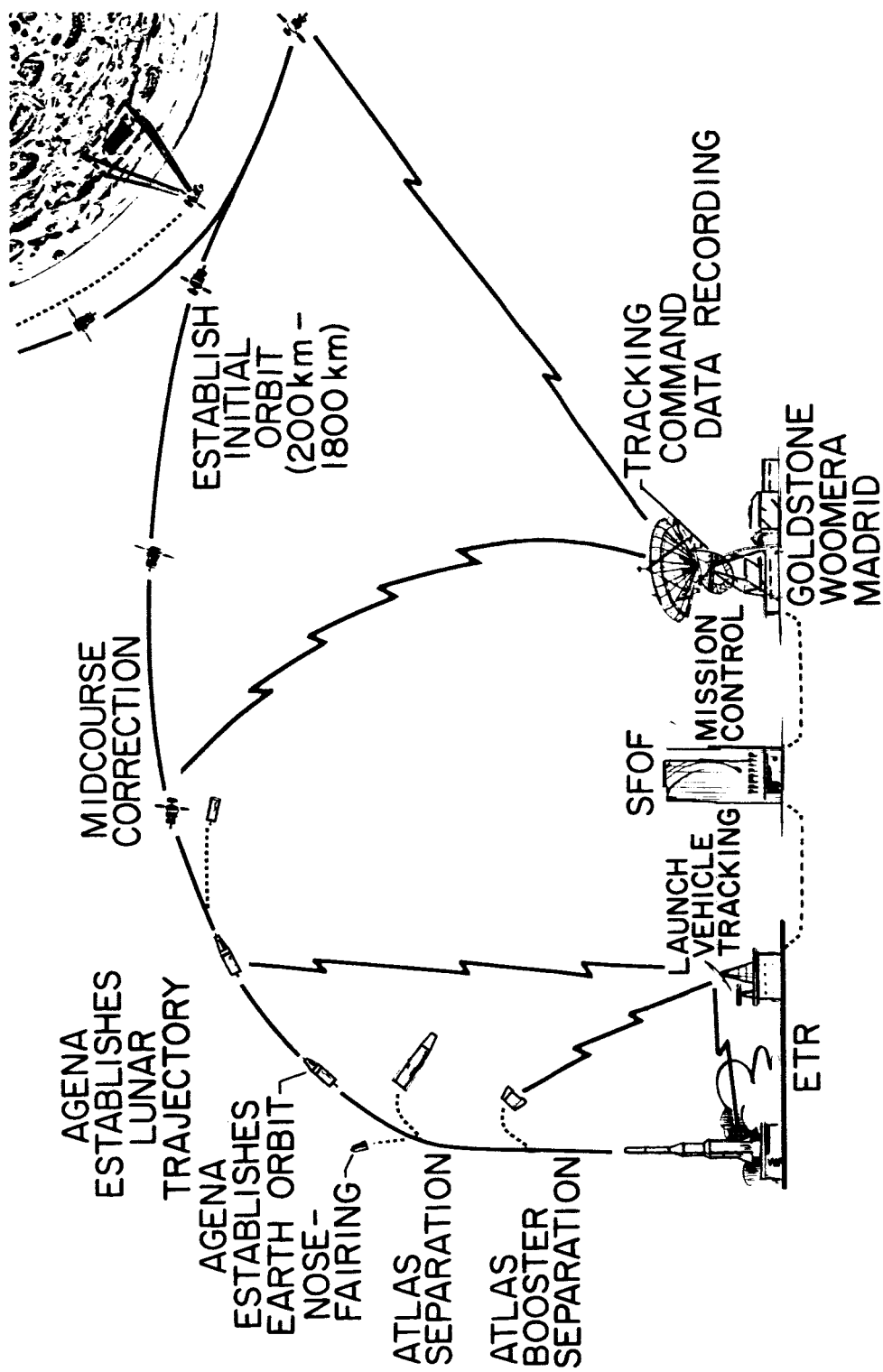
References

1. Taback, Israel: Lunar Orbiter: Its Mission and Capability. Preprint 64-7, Am. Astronaut. Soc., May 1964.
2. Taback, I.; and Brummer, E. A.: The Lunar Orbiter. Presented at AIAA Unmanned Spacecraft Meeting (Los Angeles, Calif.), Mar. 1-3, 1965.
3. Kosofsky, Leon J.; and Broome, G. Calvin: Lunar Orbiter: A Photographic Satellite. Presented at Society of Motion Picture and Television Engrs. (Los Angeles, Calif.), Mar. 28-Apr. 2, 1965.
4. Michael, William H., Jr.; and Tolson, Robert H.: The Lunar Orbiter Project Selenodesy Experiment. NASA paper presented at the Second International Symposium on the Use of Artificial Satellites for Geodesy (Athens, Greece), Apr. 27-May 1, 1965.
5. System Capabilities and Development Schedule of the Deep Space Instrumentation Facility, 1964-68; Jet Propulsion Laboratory Technical Memorandum No. 33-83, Revision 1, Apr. 24, 1964.

Table I.- Communications Link Design

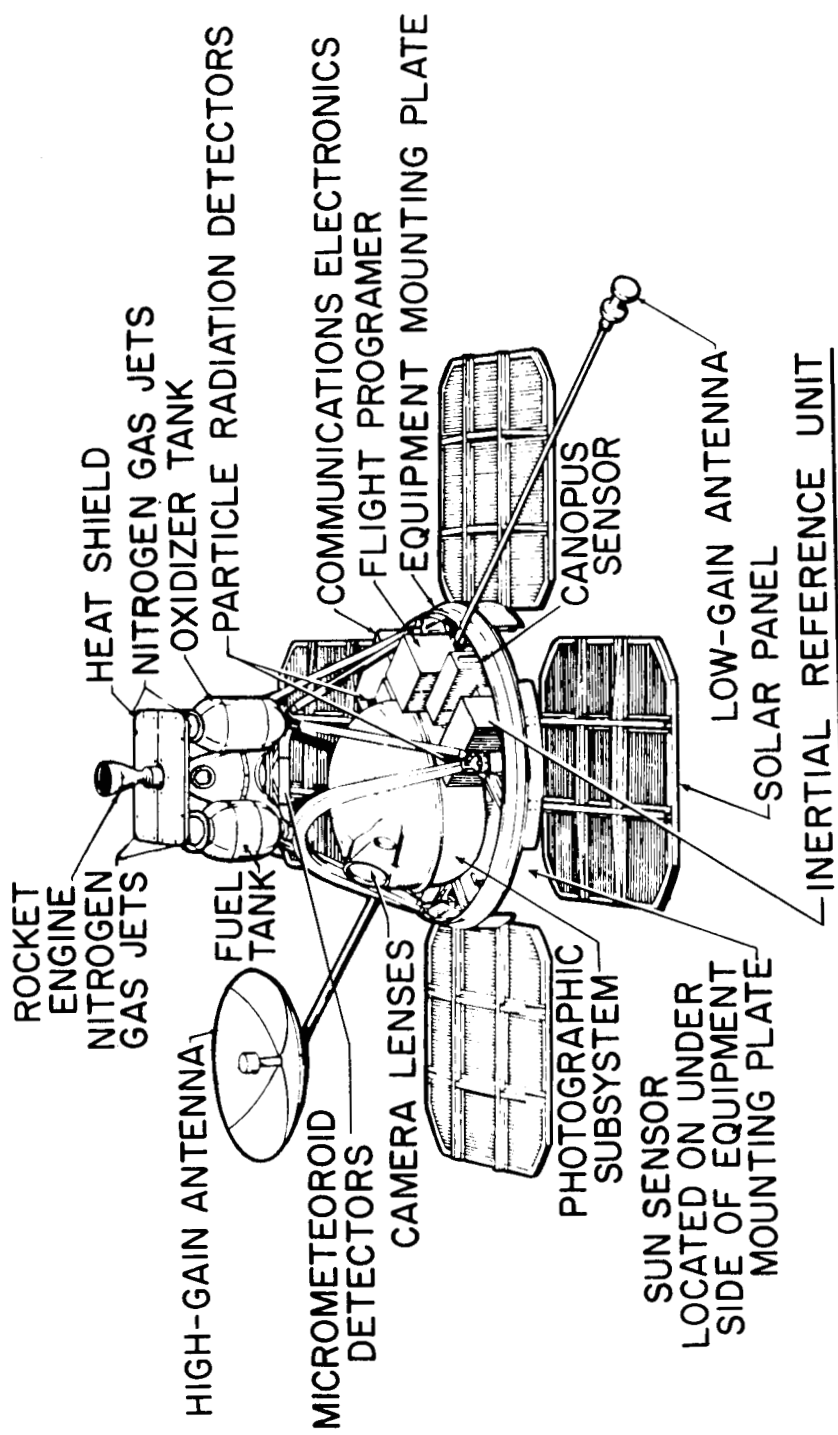
Parameter	Command	Telemetry	Video
Transmitter power	70.0 dBm	26.0 dBm	40.0 dBm
Transmitting antenna gain . . .	51.0*	-1.0	23.5
Space loss	-210.7	-211.4	-211.4
Circuit loss	-3.4	-0.7	-2.6
Receiving antenna gain	-3.0	53.0	53.0
Total received power	-96.1 dBm	-134.1 dBm	-97.5 dBm
Carrier tracking PLL:			
Modulation loss	-2.2	-5.8	
Received carrier power . . .	-98.3 dBm	-139.9 dBm	
Loop noise bandwidth	20.0	10.8	Not applicable
Threshold S/N ratio	9.0	9.0	
Threshold power	-134.3 dBm	-156.6 dBm	
Performance margin	36.0	16.7	
Data channel:			
Modulation loss	-7.9	-3.2	0
Received data power	-104.0 dBm	-137.3 dBm	-97.5 dBm
Noise bandwidth	24.0	17.0	65.2
Threshold S/N ratio	12.0	10.3	7.0
Threshold power	-127.3 dBm	-149.1 dBm	-104.2 dBm
Performance margin	23.3	11.8	6.7
Output S/N ratio	61.8 rms/rms	22.1 rms/rms	40.5 rms/rms

*All values in dB unless otherwise noted.



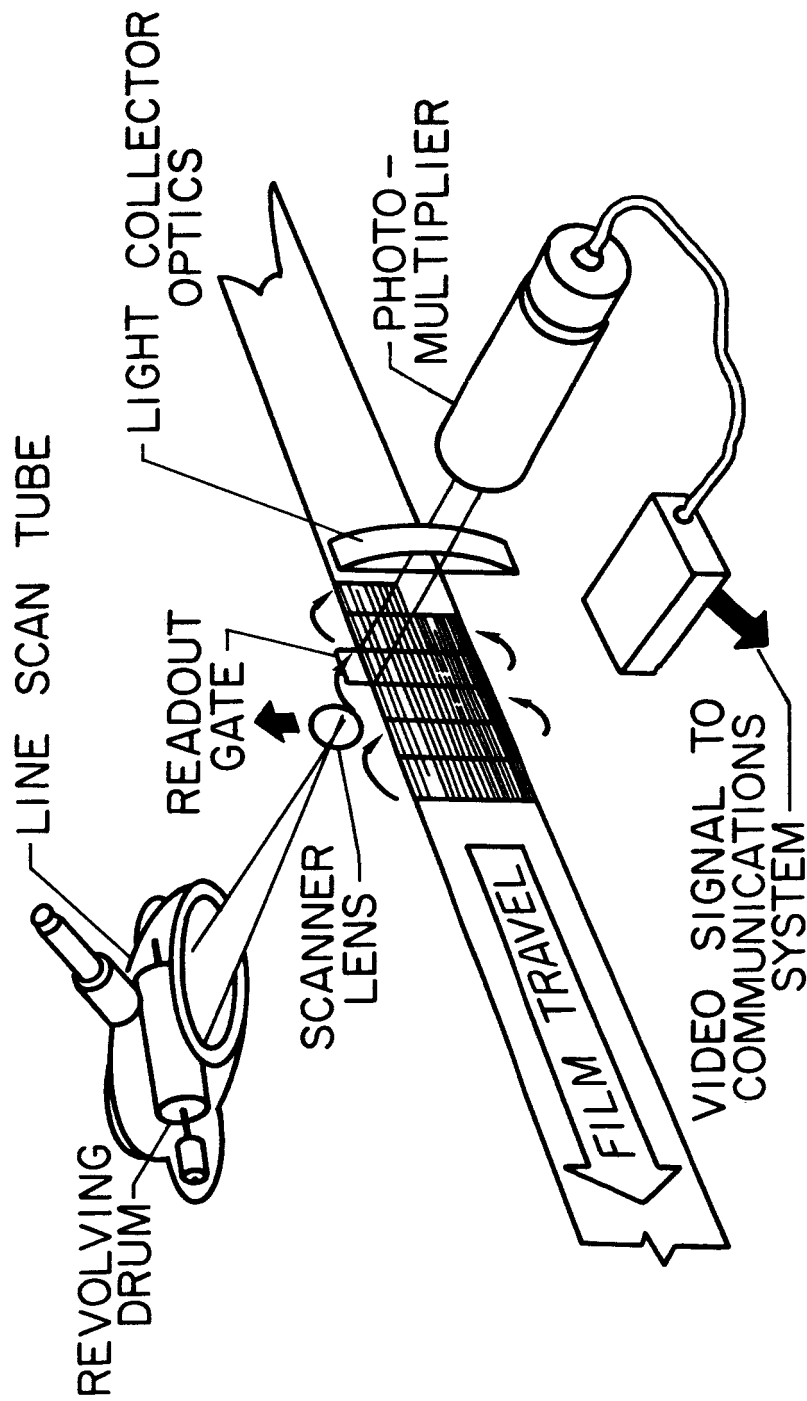
NASA

Figure 1.- Mission operations.



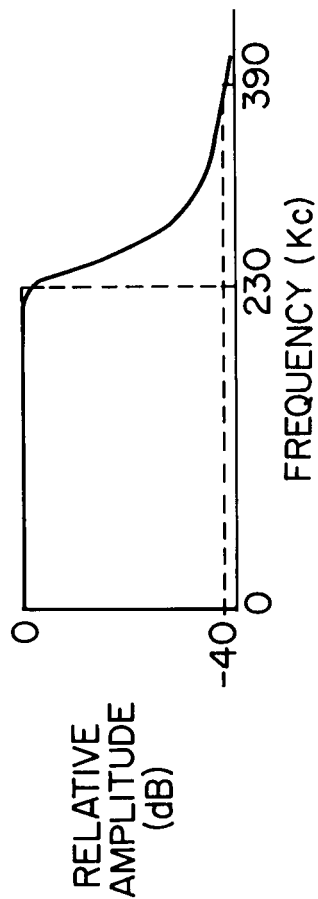
NASA

Figure 2.- Spacecraft configuration.

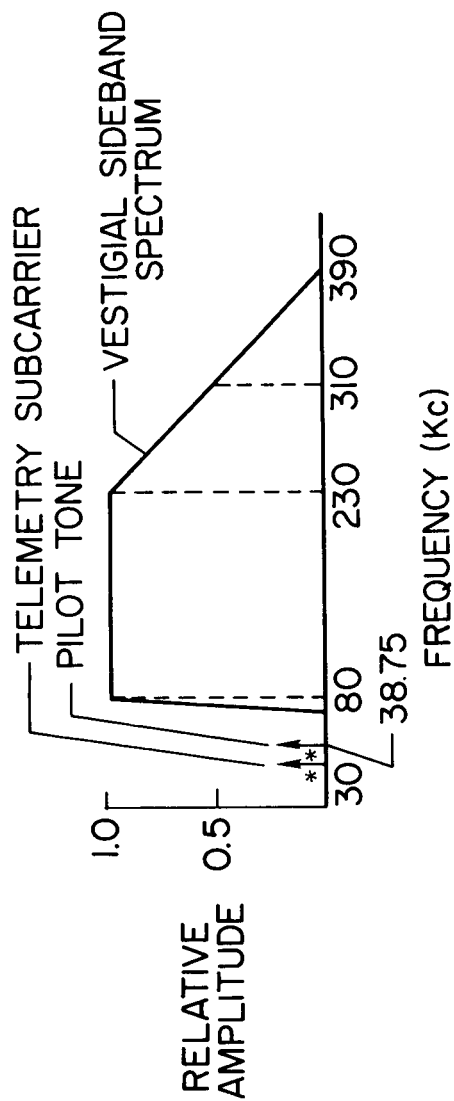


NASA

Figure 3.- Photographic readout system.

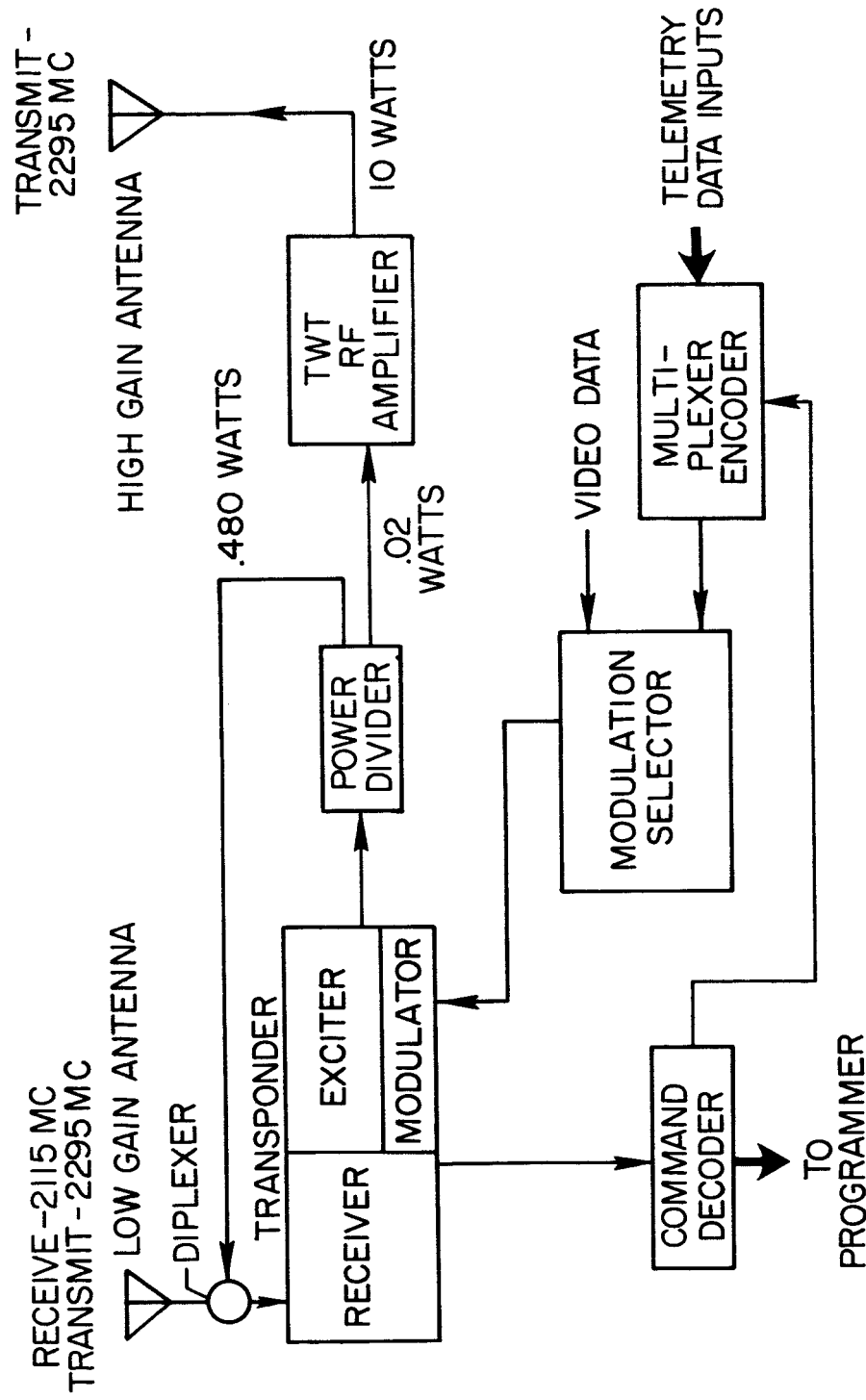


(a) Video signal.



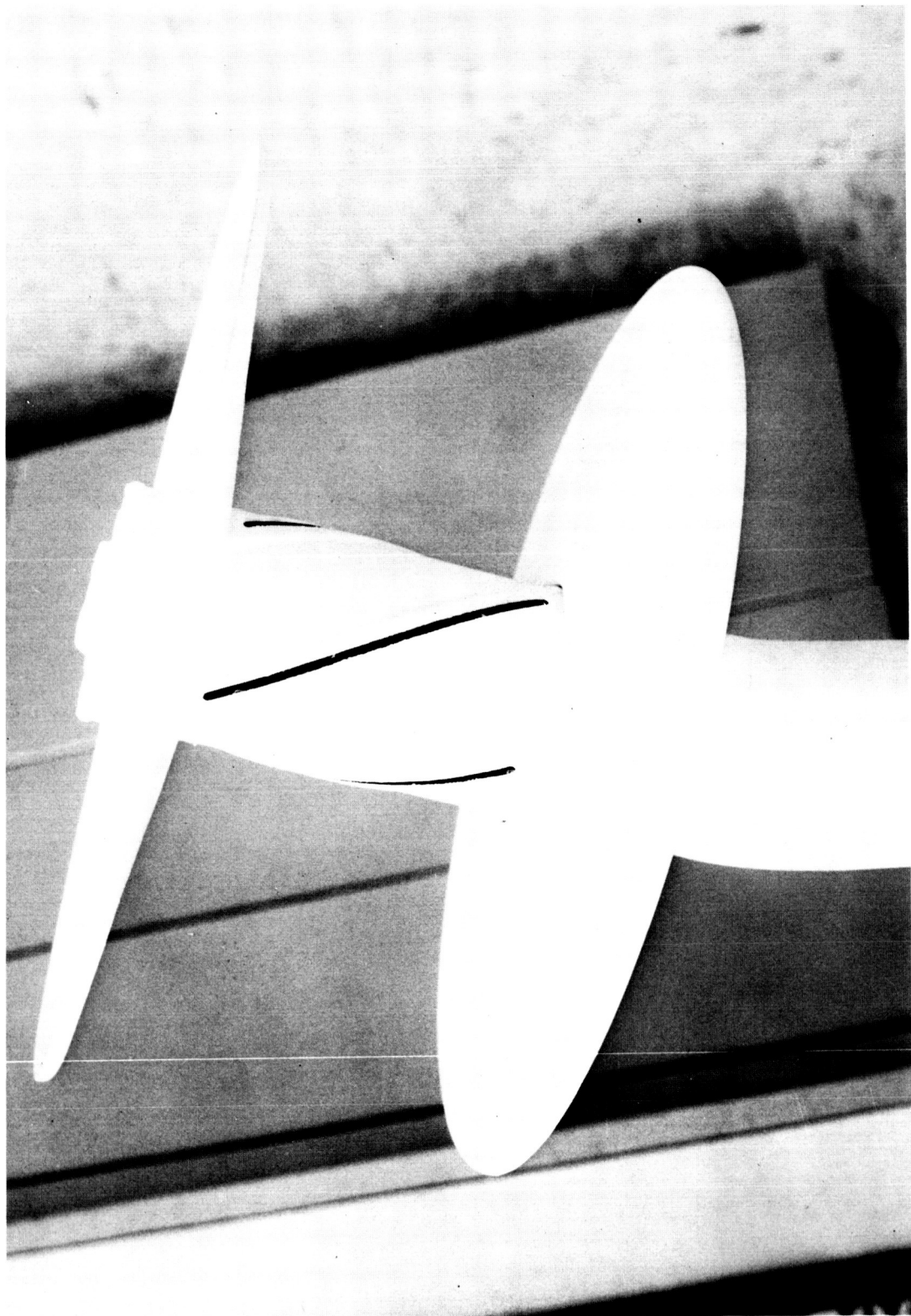
(b) Composite signal.

Figure 4.- Frequency spectrums.



NASA

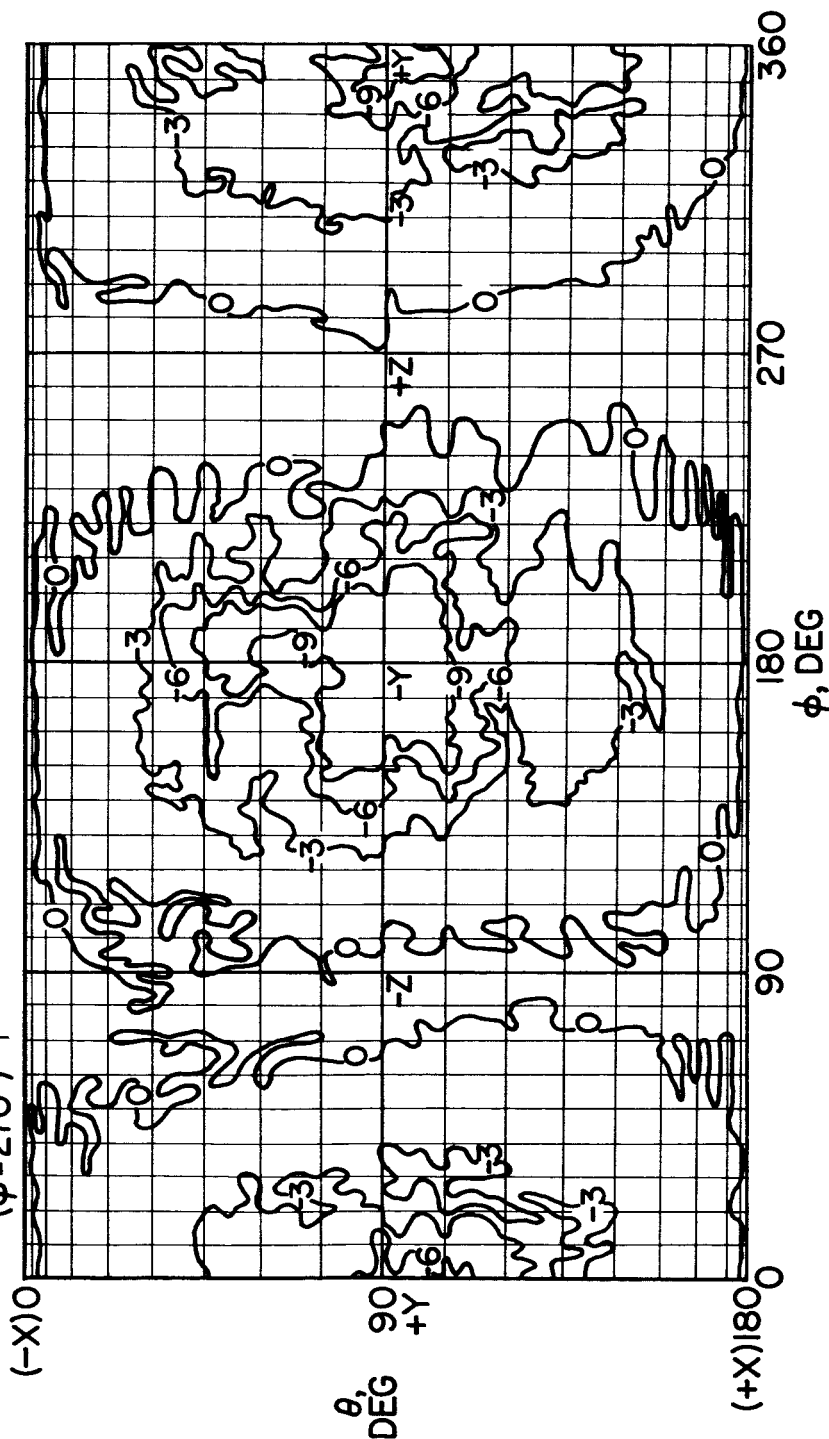
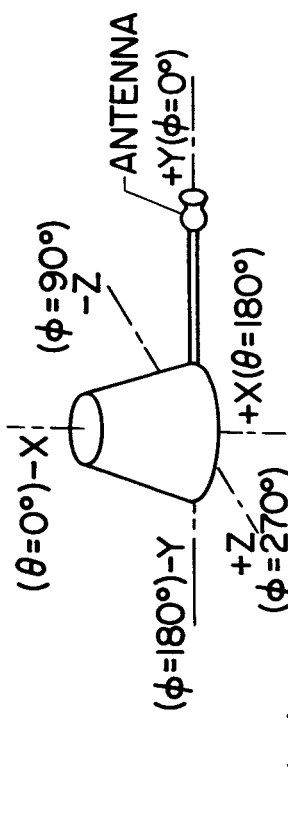
Figure 5.- Spacecraft communications system.

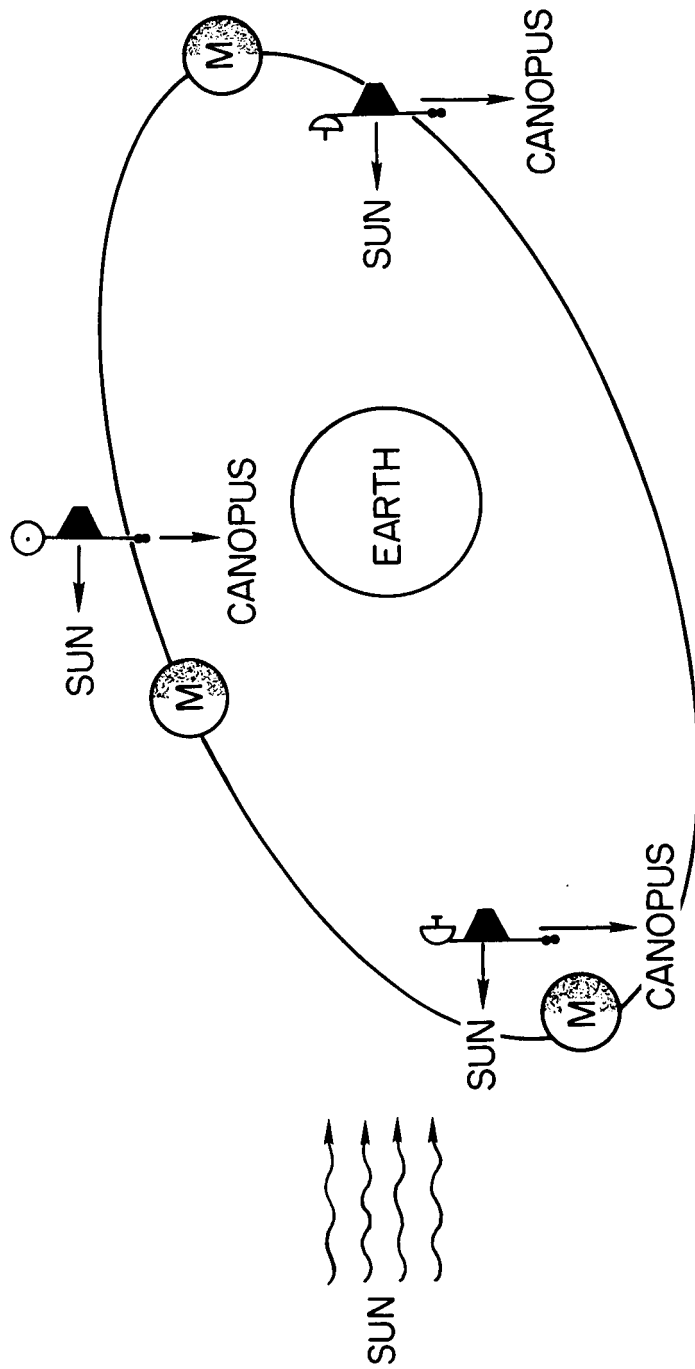


NASA

Figure 6.- Low gain antenna.

GAIN VALUES SHOWN IN THE CONTOUR
LINES ARE DB REFERRED TO AN
ISOTROPIC RADIATOR (RHC
POLARIZATION)





NASA

Figure 8.- Orientation of spacecraft antennas; rotation of high gain antenna.

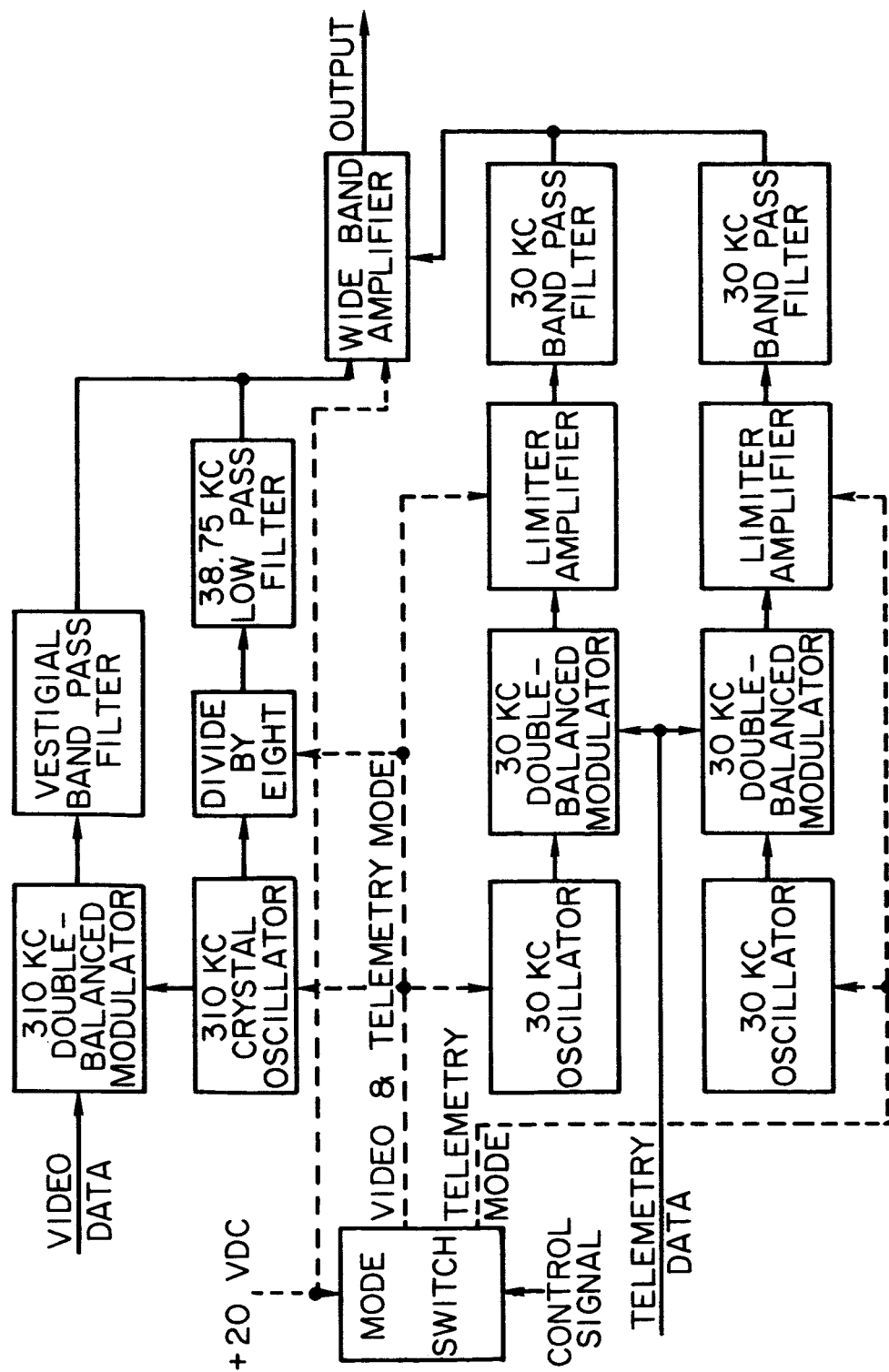
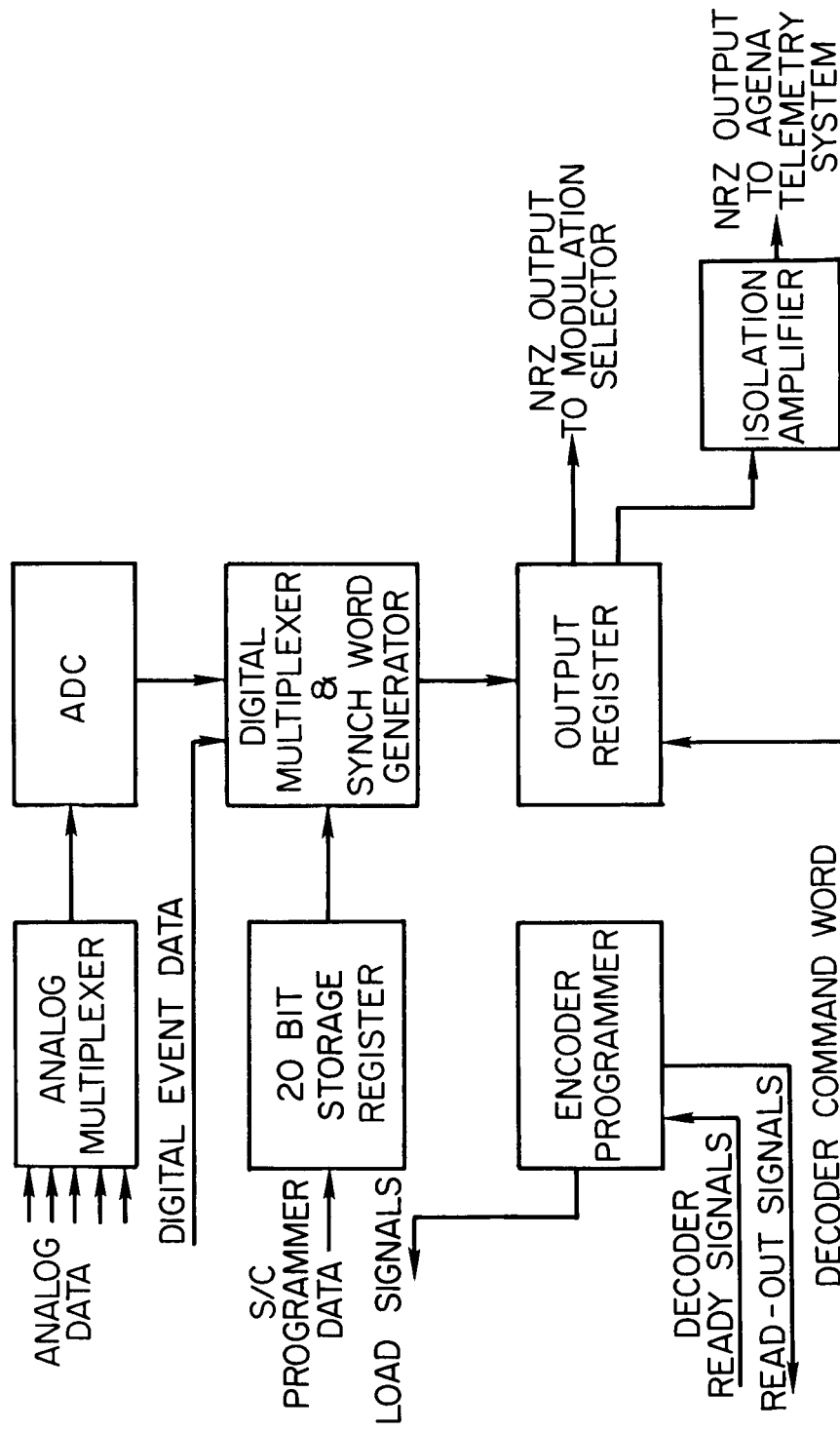
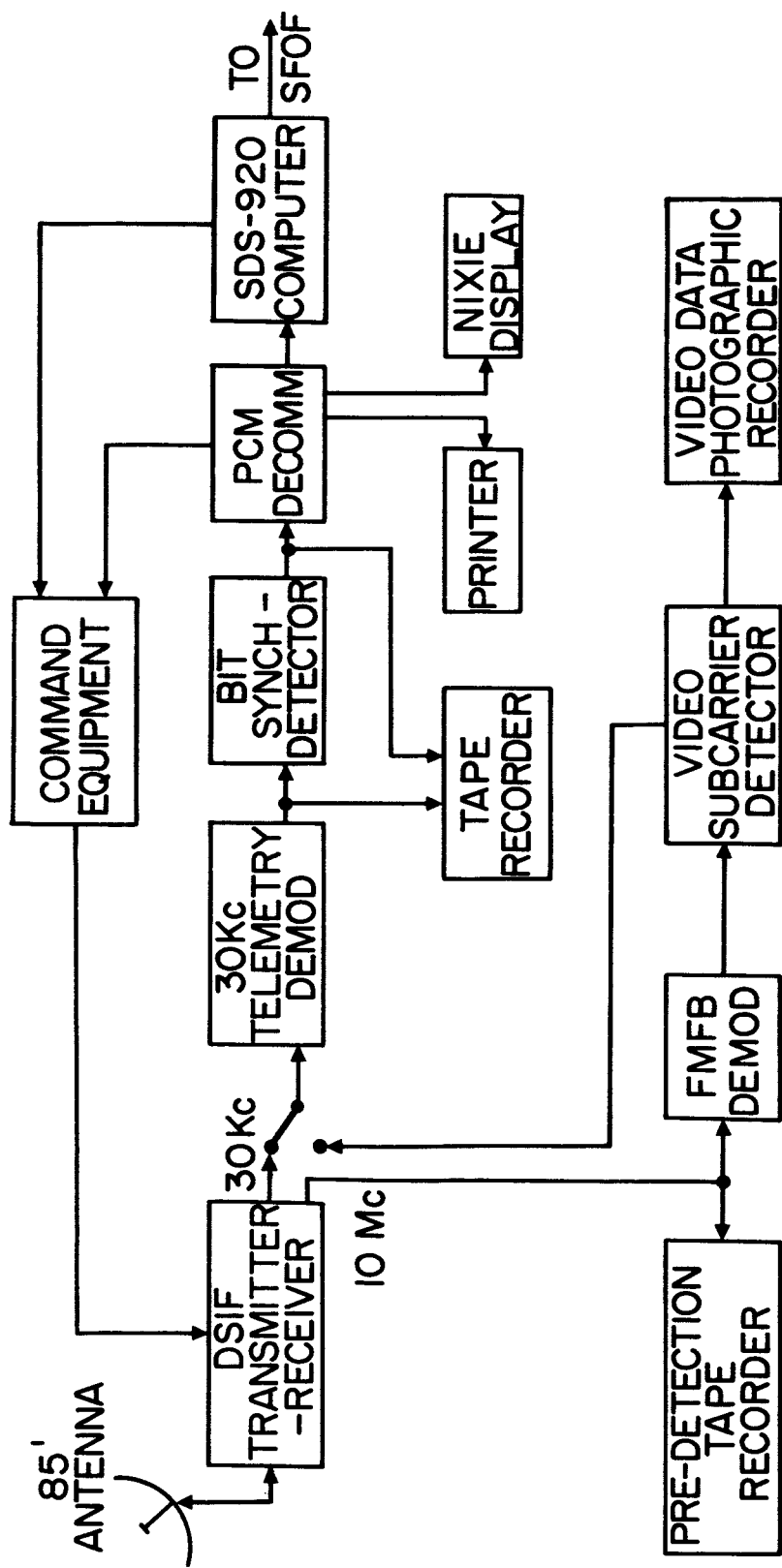


Figure 9.- Modulation selector block diagram.



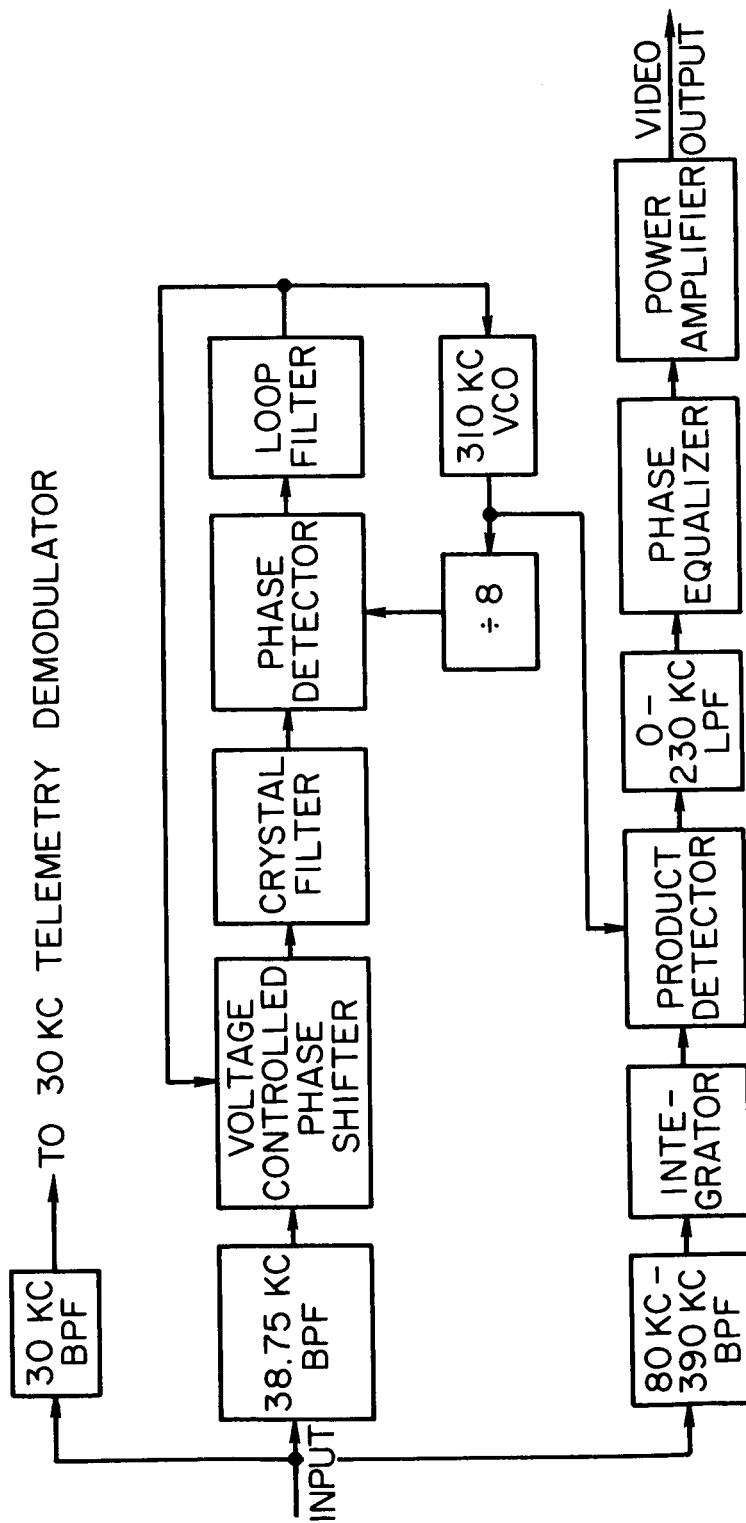
NASA

Figure 10.- Multiplexer encoder.



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Figure 11.- Ground system block diagram.



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Figure 12.- Video subcarrier detector.

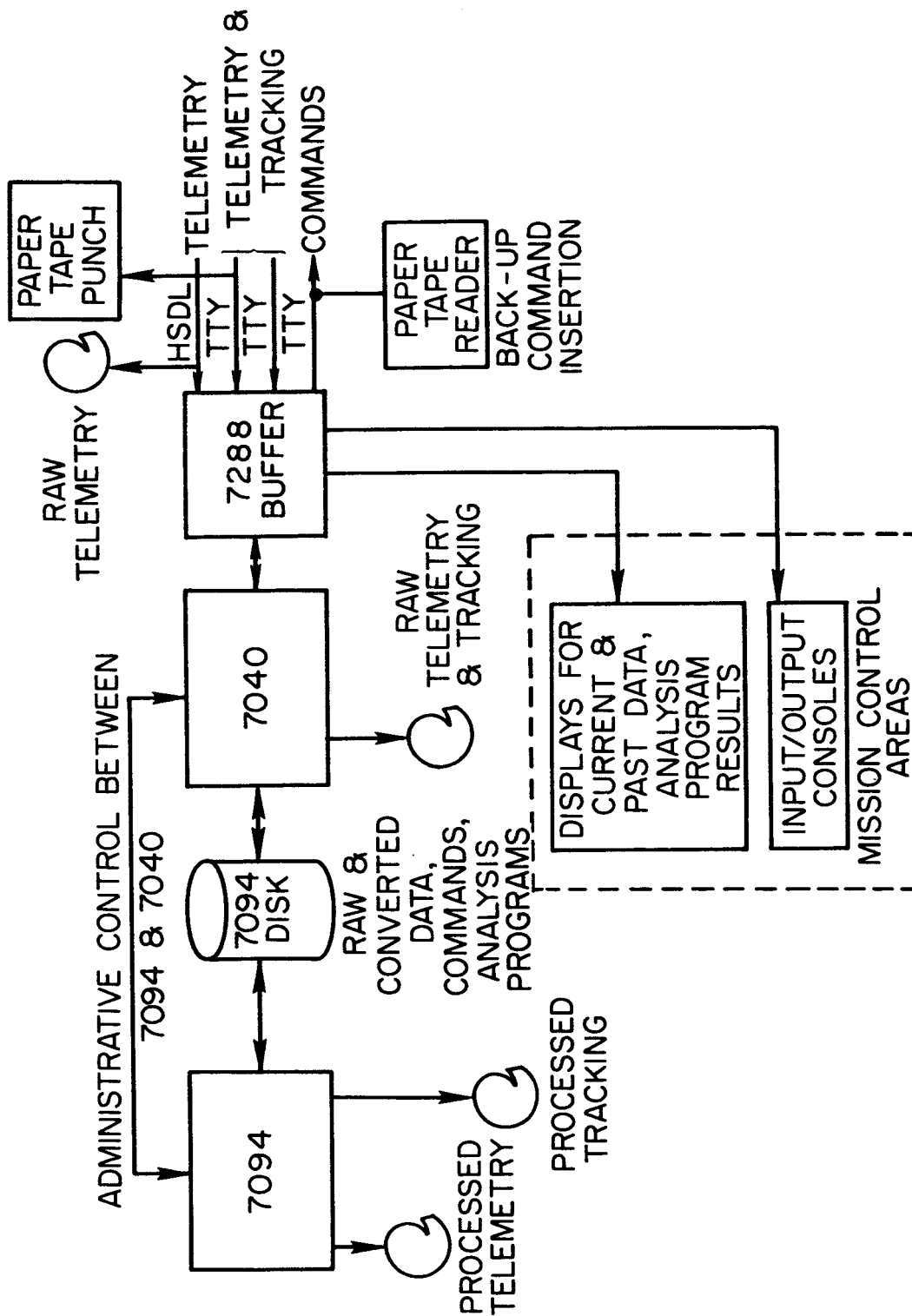


Figure 13.- SFOF data flow diagram.